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EXPERIMENTAL AEROTHERMODYNAMIC RESEARCH OF HYPERSONIC AIRCRAFT

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Ames Research Center
Moffett Field, California 94035

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16. Abstract

Wind-tunnel tests were conducted to establish a benchmark experimental data base for a generic hypersonic aircraft vehicle. Comprehensive measurements were made at a Mach ~~number~~ of 7 to give flow visualization, surface pressure, surface convective heat transfer, and flow-field pitot pressure for a delta planform all-body vehicle. The tests were conducted in the NASA/Ames 3.5-Foot Hypersonic Wind Tunnel at Reynolds numbers sufficient to give turbulent flow. Comparisons are made of the experimental results with computational solutions of the flow by an upwind parabolized Navier-Stokes code developed at Ames. Good agreement of experiment with solutions by the code is demonstrated.

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for the period
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23 August, 1990

The need for long-range, high-speed modes of transportation has generated significant interest in high-performance hypersonic aircraft. A concept known as the National Aero-Space Plane, (NASP) envisions a lifting air-breathing vehicle that can operate across the Mach number range up to orbital velocity with the added capability of conventional aircraft-type takeoffs and landings. A concerted theoretical and experimental program is underway to give proper definition to the design of NASP. The theoretical effort entails numerical solutions of the Navier-Stokes equations for perfect and real-gas flows using state-of-the-art and advanced computer codes. Because of their versatility, the codes are a valuable asset in all stages of the design process. However, comparisons with experimental data are necessary to validate the codes and to give confidence in their applicability to practical designs.

A program has been initiated at the Ames Research Center to provide these much-needed experimental aerodynamic data. Comprehensive wind-tunnel tests were made in the 3.5-Foot Hypersonic Wind Tunnel of a generic hypersonic airplane model of a type envisioned for NASP. Briefly, the model is an all-body concept having a delta planform with 75° sweepback of the leading edges, elliptical cross sections, and a sharp trailing edge. Horizontal and vertical control surfaces can be appended to the outboard tips at the trailing edge. Previous research has shown that this configuration is stable and controllable over the operational range of Mach numbers. The present tests were conducted without control surfaces.

The present Final Technical Report covers the period from the beginning of the grant on March 1, 1986, to completion on May 31, 1990. The protocol for conducting the research was established at the beginning of the grant (Cooperative Agreement No. NCC2-416). Under terms of the agreement, the Principal Investigator, Joseph W. Cleary, and the Technical Monitor,

William K. Lockman, collaborated on conducting the experimental aerodynamic research and the wind-tunnel test programs.

A major endeavor of the experimental program was to provide aerodynamic data that would be useful for computational code validation. A selected series of tests were made to give surface pressure, convective heating, shock-wave shape, surface skin-friction lines, and flow-field properties of the shock layer. Results were presented at the Third, Fourth, Sixth, and Seventh National Aero-Space Plane Technology Symposia. A list of the papers that were presented and will be published is given as an addendum. The experimental data were shown to be in good agreement with an upwind parabolized Navier-Stokes (UPS) code. Copies of NASP papers have been submitted previously as progress reports.

As a Principal Investigator, the research covered four major work areas: (1) Stress analysis and detail designs; (2) Aerothermodynamic study tasks; (3) Wind-tunnel test operations; and (4) Data analysis and report writing. Although the Principal Investigator collaborated in all areas, he was given primary responsibility for certain research aspects of the investigation. Participation in each of the four work areas will be discussed briefly. Then a more detailed report will be given from which research aspects of primary responsibility will be apparent.

The model was designed and built by Convair Corporation specifically for wind-tunnel tests at Ames. To assure integrity of the model and sting support, a complete review was made of the Convair stress analysis for the high-temperature and dynamic-pressure conditions encountered in the 3.5-Foot Hypersonic Wind Tunnel. Loads were estimated from previous wind-tunnel force tests of a smaller model and factors of safety were evaluated.

Detailed designs and stress analyses were completed of five related components essential to the wind-tunnel test program. A new circulating-water, constant-temperature heater was designed to accommodate six electronic pressure scanners for surface pressure measurements. Commercially available heaters were too large to fit in the restricted internal volume of the model.

For aerodynamic convective heating tests of the model, a new thin-skin heat-transfer cover plate was designed. A unique feature of the design was to attach a thin sheet of stainless steel of constant thickness to the cover-plate frame, thereby eliminating the need for expensive detailed machining of thin-skin areas for the thermocouples.

Because the wind-tunnel test section is not equipped for flow-field surveys, a new survey platform was designed. The platform can be used to position pitot-pressure rakes (or other type probes) within the (x, y, z) coordinate system of the model flow field. A new flow-field and boundary-layer rake was also designed. The rake consists of multiple stainless steel probes closely spaced in the boundary-layer region and more widely spaced in the outer shock layer. The probes were flattened to give better definition to the boundary-layer profiles. In addition to the foregoing components, a 5° sideslip wedge was designed to yaw the model. All of the foregoing components were built and used in the all-body wind-tunnel test program. Furthermore, most of these components are believed useable for other future wind-tunnel test programs.

In the area of aerothermodynamic study tasks, equations were derived for hypersonic Newtonian flow over the elliptical-forebody and -afterbody of the all-body model. The equations were extended and a computer code was developed (jointly with the Technical Monitor) to give tangent-cone solutions of the flow. The solutions show the effects of angle of attack, angle of sideslip, Mach number, and specific-heat ratio on spanwise pressure distributions. Comparisons were made with experimental results and UPS computational solutions of the flow in a previously mentioned NASP document.

Before beginning the pressure distribution tests, a brief analysis was made of unpublished pressure distributions from tests of the model in the Unitary Plan Wind-Tunnel Facilities. The tests were made at Mach numbers of 2.5, 3.0, and 3.4. The data were analyzed to show the effects of angle of attack, nose bluntness, Mach number, and forebody/afterbody juncture on windward and leeward spanwise-pressure distributions. The results provided a basis for analyzing these same effects at hypersonic Mach numbers.

A simplified inviscid model of the forebody and afterbody flow-field was developed to demonstrate flow-field structure on model centerline. Because the forebody of the model is an elliptical cone, the bow-shock wave is basically conical as is the entire forebody shock layer except for the boundary layer. For such flows, an approximation using the known wave angle in the angle-of-attack plane was previously published in NASA TN D-5951 (by the Principal Investigator). The method yields accurate inviscid flow-field estimates for various conical bodies. The afterbody flow-field for the all-body model is quite complex; it is composed of an outer conical region, a central expansion region, and a viscous region at the wall. Using the above approximate method, a simplified inviscid model of this flow-field was developed and compared with experiment. The main features of the centerline flow-field structure compare favorably with experiment for a range of angles of attack. In the work area of wind-tunnel operations, the Principal Investigator assumed a hands on position for the numerous tasks involved. Briefly, the wind-tunnel tests required model preparation and installation, data acquisition, and data reduction. Calibration and instrumentation checks were made before and after a wind-tunnel run to assure valid data. Each of the various tes spanned a duration of one to two (or more) months.

The final work area, data analysis, and report writing, can be time consuming. Large amounts of data were produced during the wind-tunnel test program. Test results were published in a timely manner in the four previously mentioned NASP Technology Symposia papers. Moreover, an AIAA paper entitled "Experimental and Computational Surface and Flow-Field Results for an All-Body Hypersonic Aircraft" will be given in at the AIAA 8th Applied Aerodynamics Conference in Aug. 1990. The paper gives detailed comparisons of flow visualization, surface pressure, heating, and shock-layer pitot-pressure measurements with Navier-Stokes solutions of the flow from the Ames UPS code. Generally good agreement is shown by the comparisons. The experimental results from this study are invaluable for the validation and or calibration of the UPS code and other codes being developed at Ames and by other organizations.

With regard to specific aspects of the research program, the Principal Investigator was given primary responsibility for the flow-field area of research. Detailed pitot-pressure measurements were made of the afterbody flow-field of the all-body model. It is noteworthy that turbulent flow over the afterbody is quite complex and affords a realistic test case for the capabilities of the UPS code. In addition to pitot-pressure, measurements of bow shock-wave shape and surface oil-flow patterns were essential to the flow-field analysis. Tests were completed for the sharp-nose model. Effort was expended in preparation for tests to determine the effects of nose bluntness. However, wind-tunnel scheduling conflicts precluded tests of the effects of bluntness.

At present, a final report is being prepared and near completion on the all-body sharp-nose flow-field data. Completed results of the afterbody pitot-pressure flow-field tests will be given in tabulated form for centerline and outboard (lateral) survey stations. In addition, the report will have figures that show the effects of angle of attack and longitudinal station on pitot-pressure distributions. Since the boundary layer is quite thick and the rake probes are closely spaced, detailed boundary-layer profiles are shown for complex turbulent flows.

Brief excerpts from the final report are given now to show the more interesting aspects of the flow-field investigation. In figure 1, the model is shown mounted on the sting support in the wind-tunnel. The model is instrumented with centerline and outboard cantilever mounted rakes which can be located at various longitudinal positions. A shadowgraph of the flow-field is shown in figure 2 for $\alpha = 5°$. Pertinent features of the flow on the windward side are the bow-shock wave, an expansion of the flow at the forebody/afterbody juncture, and the turbulent boundary layer on the forebody. A thick turbulent boundary layer on the afterbody is barely visible. The model is shown with the alternate pitot-pressure rake that was used in preliminary tests. The boundary-layer and flow-field rakes shown in figure 1 were used for the main investigation.

A comparison of centerline pitot-pressure data with an inviscid model of the flow-field is shown in figure 3. This figure demonstrates the structure of the afterbody flow-field. Since the bow-shock wave is straight, the flow-field has a conical-flow region adjacent to the wave. The expansion of the flow at the forebody/afterbody juncture is similar to the Prandtl-Meyer expansion of the flow model. The expansion turns the flow parallel to the afterbody boundary layer, which is represented as a two-dimensional flow (planar) region by the flow-field model.

The effects of x/L (rake station) on pitot-pressure distributions for $\alpha = 0°$ are shown in figure 4. At each station, the distributions are composed of conical, expansion, and viscous regions of the flow-field as previously discussed. The edge of the boundary layer (viscous region) was estimated from shadowgraphs. It can be seen that the relative thicknesses of the flow-field regions are highly dependent on x/L. Just after the forebody/afterbody juncture (x/L = 0.70), the expansion and the thick turbulent boundary layer are closely coupled. Further aft (x/L ≥ 0.80), it is hypothesized that the expansion terminates external to the boundary layer. The somewhat abrupt change in pitot-pressure near the boundary layer edge (figure 4) at x/L = 0.90 is believed to be a manifestation of the phenomenon.

The effects of varying angle of attack on pitot-pressure distributions are shown in figure 5. Similar to $\alpha = 0°$, the distributions on the windward side are composed of conical, expansion, and viscous regions. Increasing angle of attack has a significant effect on the juncture expansion because the forebody Mach number, which modulates the expansion, is reduced. On the leeward side, the distributions for $\alpha = -5°$ and $-10°$ appear related as a family of curves to those on the windward side.

Pitot-pressure distributions at the outboard station, y/L = 0.167, are shown in figure 6 for $\alpha = 0°$. Previous results on centerline are shown for comparison. The outboard distributions, similar to centerline, have conical, expansion, and viscous regions of the flow-field. However, for the same distance from the surface, pitot-pressure is greater outboard than on centerline. In summary, a comprehensive wind-tunnel investigation was conducted for a generic airplane model in support of NASP technology development. Tests were made to give surface pressure, heating, shock-wave shape, surface skin-friction lines, and flow-field pitot pressure. The experimental data were found to be in good agreement with the theoretical UPS code. Results were presented at NASP Symposia and will be given at the AIAA Applied Aerodynamics Conference.

A list of the publications from this investigation is presented here.

1. Lockman, William K.; Lawrence, Scott L.; and Cleary, Joseph W.: Flow-Visualization Results for an All-Body Hypersonic Aircraft. Third National Aero-Space Plane Technology Symposium, Paper No. 6, June 1987.

2. Lockman, William K.; Cleary, Joseph W., and Lawrence, Scott L.: Flow Visualization and Pressure Distributions for an All-Body Hypersonic Aircraft. Fourth National Aero-Space Plane Technology Symposium, Paper No. 53, Feb. 1988.

3. Lockman, William K.; Cleary, Joseph W.;and Lawrence, Scott L.: Experimental and Computational Flow-Field Results for an All-Body Hypersonic Aircraft. Sixth National Aero-Space Plane Technology Symposium, Paper No. 37, April 1989.

4. Lockman, William K.; Lawrence, Scott L.; and Cleary, Joseph W.: Surface Heat Transfer and Pitot-Pressure Survey Results for an All-Body Hypersonic Aircraft. Seventh National Aero-Space Plane Technology Symposium, Paper No. 6, Oct. 1989

5. Lockman, William K.; Lawrence, Scott L. and Cleary, Joseph W.: Experimental and Computational Surface and Flow-Field Results for an All-Body Hypersonic Aircraft. AIAA Paper No. 90-3067, Aug. 1990.

6. Cleary, Joseph W. and Lockman, William K.: Experimental Flow-Field Results for an All-Body Hypersonic Aircraft Model. Proposed NASA TM, 1990.

7. Cleary, Joseph W. and Lockman, William K.: Newtonian and Tangent-Cone Approximations of the Spanwise Pressure Distributions for an All-Body Hypersonic Aircraft. Proposed NASA TM, 1990.

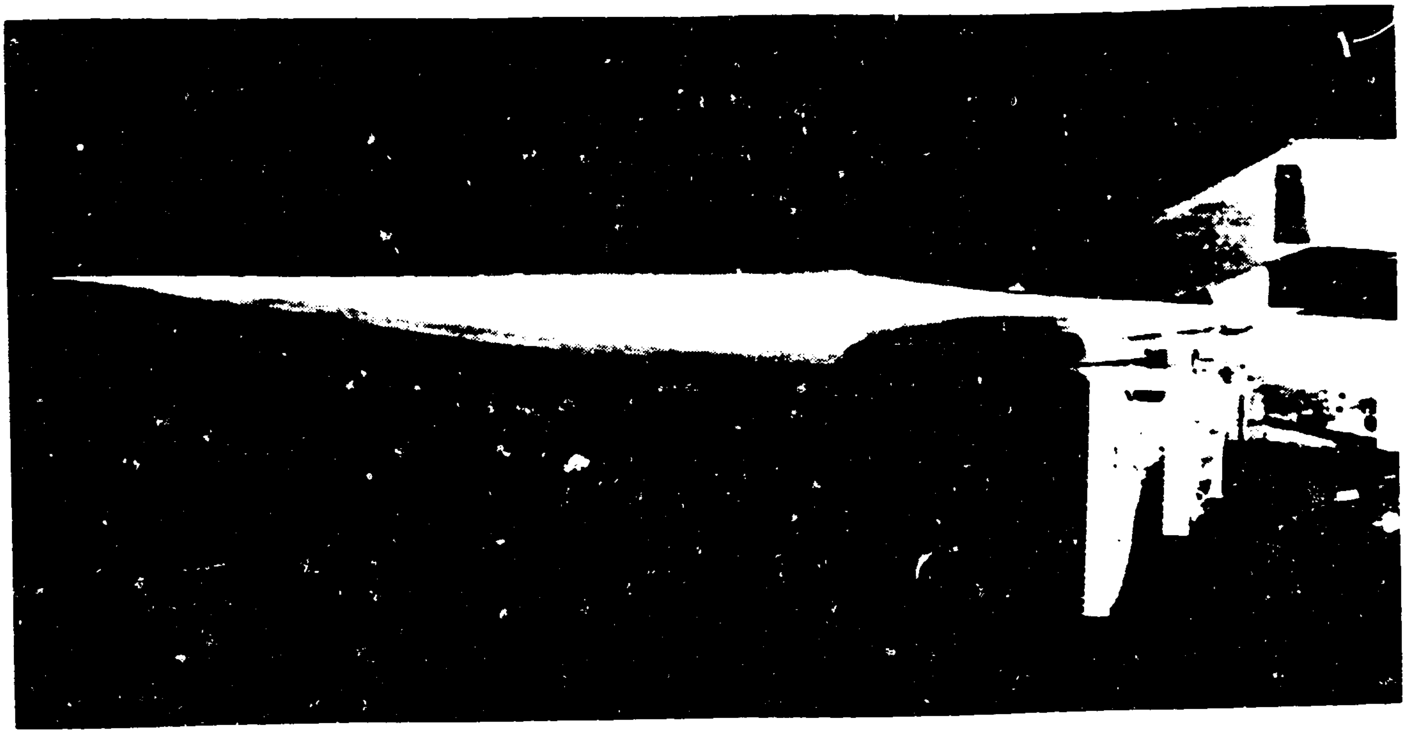

Fig. 1 Photograph of pitot-pressure survey setup with rakes on centerline and outboard.

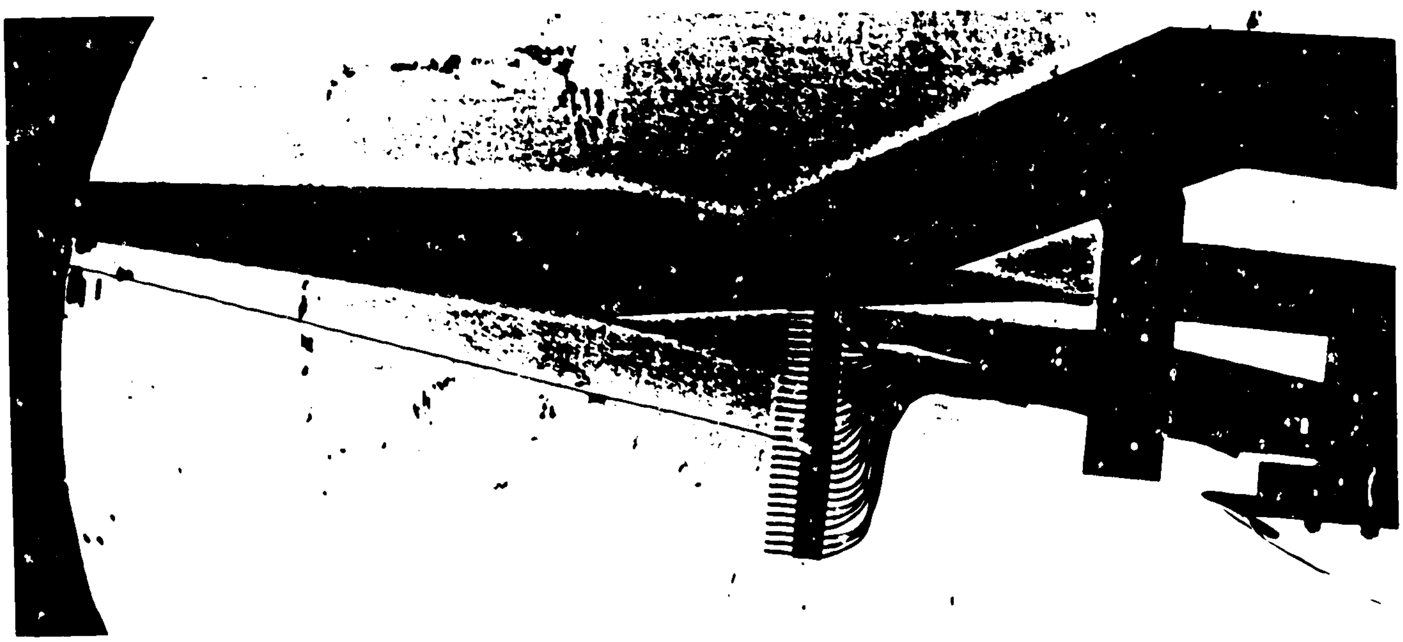

Fig. 2 Shadowgraph of flow field: $\alpha = 5°$, $M_\infty = 7.4$, $Re_{\infty,L} = 15 \times 10^6$.

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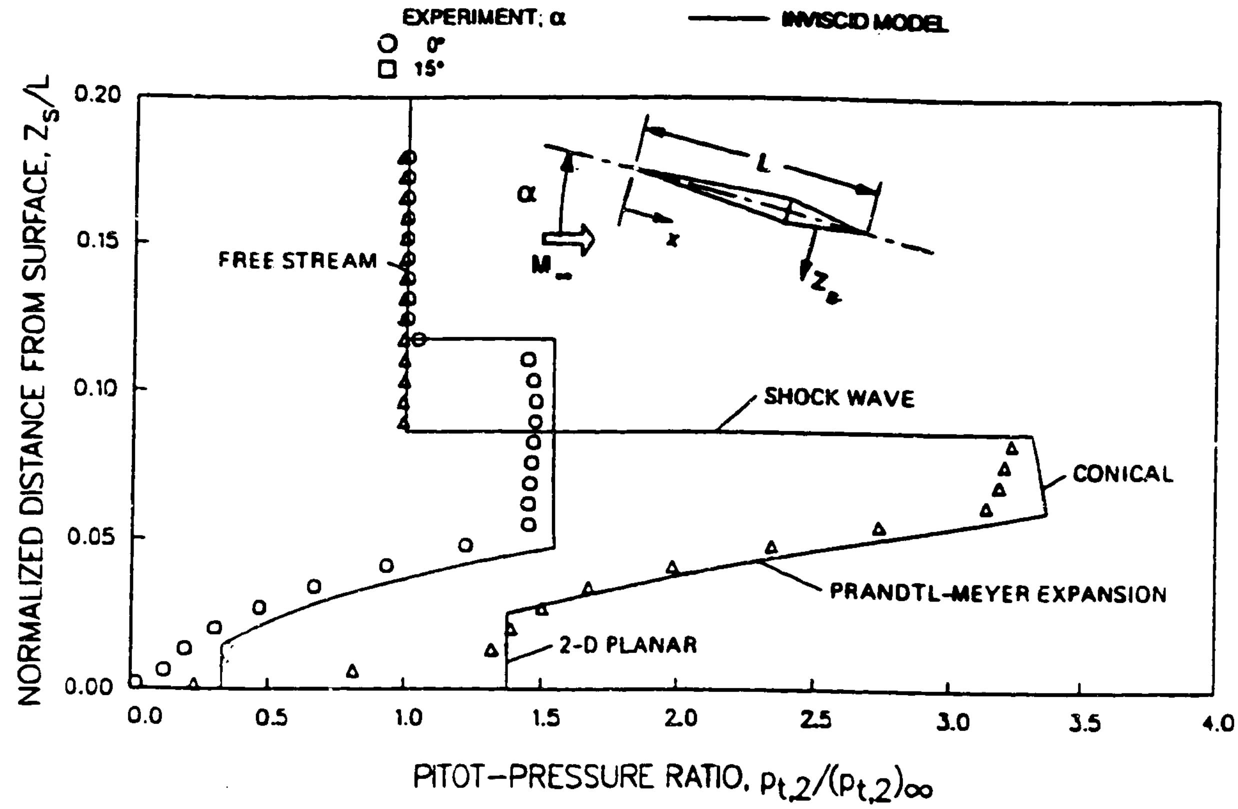

Fig. 3 Comparison of centerline pitot pressure with inviscid model of flow; $M_\infty = 7.4$.

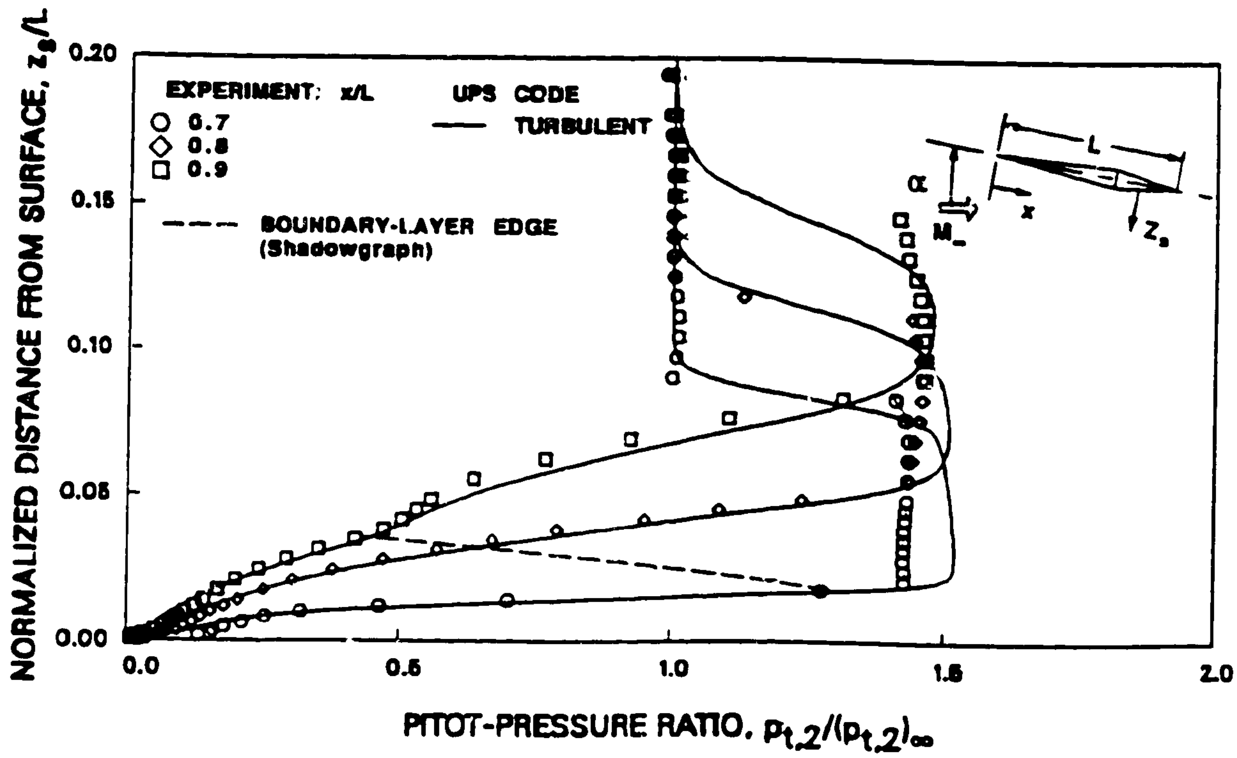

Fig. 4 Effect of rake station on pitot-pressure distributions; $\alpha = 0°$, $M_\infty = 7.4$, $Re_{\infty,L} = 15 \times 10^6$.

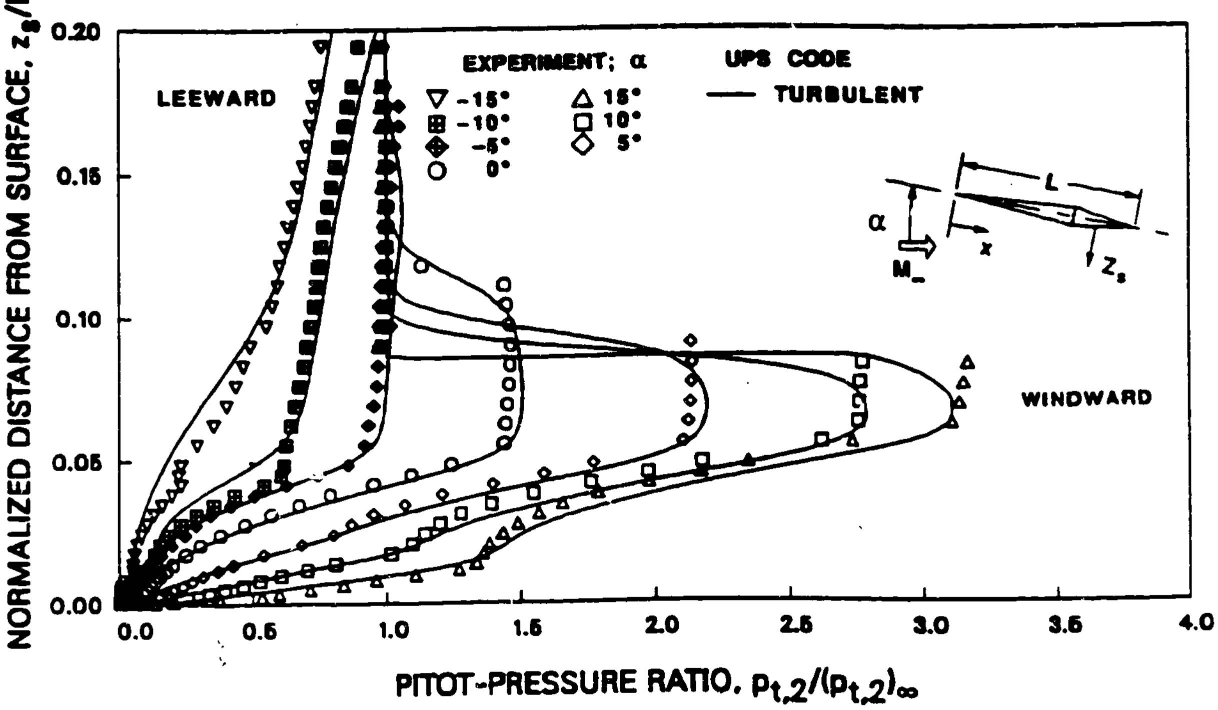

Fig. 5 Effect of angle of attack on pitot-pressure distributions; M_∞ = 7.4, $Re_{\infty,L}$ = 15 x 10^6.

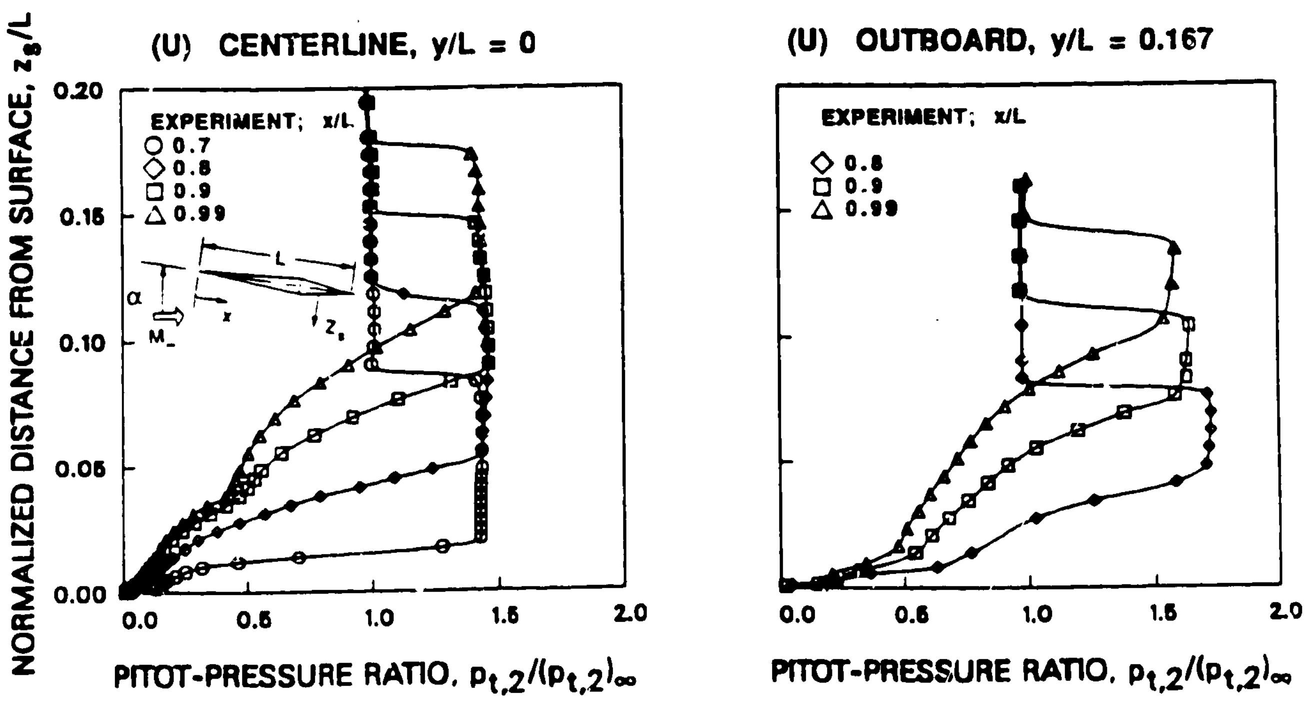

Fig. 6 Comparison of pitot-pressure distributions at centerline and outboard stations; M_∞ = 7.4, $Re_{\infty,L}$ = 15 x 10^6.

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